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FLIGHT TESTS AND EARLY MISSIONS FOR

ELECTRIC PROPULSION SYSTEMS

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INTRODUCTION

This paper will describe the NASA programs for flight tests of electric thrusters and for early space missions using electric propulsion. The flight tests described herein are scheduled for the time period 1963 to 1970.

The electric thruster flight tests during this period will have three main objectives:

- (1) To supplement vacuum tank tests of electric thrusters and provide data on phenomena for which tank tests are inadequate or questionable.
- (2) To provide data on problems involved in the integration of thrusters with power supplies and flight vehicles.
- (3) To qualify electric thrusters for mission applications.

SPACE ELECTRIC ROCKET TESTS (SERT)

SERT I

Two ballistic flights using four-stage Scout vehicles are planned during the time period indicated in figure 1. On each of these flights, two ion engines will be operated. The payload package is shown in figures 2 and 3. It will weigh about 350 pounds. Flight time above 200 n. miles will be about 50 minutes, and during this time useful ion engine performance data can be obtained.

Purpose. The primary objective of the SERT I flights is to establish the feasibility of neutralization of ion beams in space. Vacuum tank tests indicate that neutralization is being effected by electron emitters located at the back of ion engines, but certain questions exist regarding tank data.

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One question stems from the effect of the vacuum tank walls. During continuous operation of an ion rocket the ion beam provides a conducting path for movement of electrons from the tank wall to any point within the beam where an unbalance in charge may exist. Adding to the problem is the fact that the high-velocity ions have ample energy to produce secondary electrons from the tank walls, target, or wherever the ion beam impinges. Reference 1 indicates that the current of secondary electrons can constitute an appreciable fraction of the ion beam current, especially for surfaces with adsorbed gases such as would be present in existing test facilities.

An additional effect of tank walls is the capacitance between the ion beam and the walls. For tanks having small dimensions normal to the ion beam, this capacitance can result in induced charges in the ion beam boundaries which might contribute toward beam neutralization.

Another question regarding the validity of vacuum tank tests is the effect of the dilute plasma which may fill such tanks. Electrical breakdown has been observed across large gaps and with low potential differences, indicating the presence of such a plasma (reference 2).

The technique for pulsed operation of ion rockets has recently been developed by Sellen and Shelton of Space Technology Laboratories (ref. 3) and has served to eliminate the continuous ion beam between the engine and the beam impact point. Using the pulsed-beam technique, neutral ion beams are still obtained when suitable means of electron addition are employed in the neutralizer system. The pulsed-beam testing technique is generally considered to have removed the major source of doubt concerning vacuum tank testing. However, for the reasons outlined above, some doubts regarding vacuum tank tests still remain. Accordingly, flight tests of ion engines are considered essential to make certain that ion rocket beams can be neutralized in space.

Engines. - The two engines to be tested in the two SERT-I flights have basically different neutralizer systems. Both of these neutralizer systems have demonstrated effectiveness in producing neutral beams during both steady and pulsed operation of these engines in vacuum tanks. One of these engines was developed at the NASA Lewis Research Center (ref. 4) and the other was developed at Hughes Research Laboratories under NASA contract (ref. 5). The Lewis engine employs electron bombardment of mercury vapor to produce its ions; a flight prototype model of this engine is shown in figure 4. The Hughes engine is mounted in a small vacuum pod during launch. This permits prolonged heating and outgassing of the propellant supply lines and the porous ion emitter.

CASE FILE COPY

The major engine characteristics and operating conditions are as follows:

	<u>Lewis</u> <u>Engine</u>	<u>Hughes</u> <u>Engine</u>
Ion beam outside diameter, in.	4	3
Overall engine diameter, in.	7.5	4
Engine weight (including mount), lbs.	9.3	14
Propellant	Mercury	Cesium
Specific impulse, sec.	5000	9000
Total input power, watts	1400	610
Power efficiency, percent	50	51
Propellant utilization efficiency, percent	80	96
Thrust, millipounds	6.4	1.6
Beam current, MA	280	59
Current density, MA/CM ²	6.9	3.5
Beam perveance, micropervs	2.24	0.15

The Lewis engine (fig. 4) employs the simplest neutralizer system, and it consists of a heated tantalum strip immersed in the ion beam. A virtual anode forms at about the distance from the emitter where the emitted electron density equals the ion density. The acceleration distance for the electrons with space-charge-limited flow can thus be a small fraction of a millimeter, permitting very small accelerating potentials to be used. An immersed emitter is subject to sputtering erosion, and some other system may therefore be preferred for practical ion engines. However, it should not be inferred that the immersed emitter is totally impractical, since careful design can probably keep the erosion rate to well below one percent of the propellant flow rate.

The Hughes engine (fig. 5) employs an electron trap and emitters which are adjacent to, but not immersed in, the ion beam. The portion of the beam enclosed inside the electron trap is at a positive potential relative to the remainder of the beam. This region is therefore a potential well in which electrons can be trapped for a short dwell time during which they lose some of their energy. The electron trap also serves as an accelerate-decelerate system for introduction of electrons into the beam.

A second neutralization system will also be studied during operation of the Hughes engine. The electron trap voltage will be reduced to zero, and electrons will be added by emitters arranged just outside the beam periphery and down-beam from the trap electrode.

Kaufman (ref. 2) points out that the mechanisms involved in the operation of all of these neutralizers have not been adequately explained. He speculates that plasma wave phenomena are important in the reduction in electron energy (velocity) necessary to produce beam neutrality.

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Test Techniques. - The payload package will be spin stabilized at about 170 rpm. The two engines will be operated sequentially and are arranged to produce moments about the spin axis. Thus, changes in spin rate over a period of time will permit computation of ion engine thrust; thrust data can be obtained to an accuracy level of around three percent by this method. An instantaneous measurement of spin rate will be obtained from a "vibrating string" accelerometer, which will be mounted with its sensitive axis normal to the spin axis so as to permit measurement of the centrifugal force. In addition, spin rate will be measured by a sun sensor.

The beam spreading rate will be determined during operation of the Lewis engine by means of a 5-sensor hot-wire calorimeter like the one described in reference 6. The calorimeter will be swung through the beam to measure the distribution of energy density about the beam centerline.

Any departure from normal beam spreading will also be indicated during operation of the Hughes engine of small Faraday cups located just outside the normal beam boundaries.

A rotating vane electric field meter will indicate any difference in potential between the payload package and surrounding space.

On both engines, measurements will be made of the various electrode voltages and currents. The voltages will be varied by a mechanical programmer contained in the payload package in order to start the engine, conduct the desired test sequence, and stop the engines. During operation of the Hughes engine, a command override may be employed to adjust the voltage on the electron trap electrode.

The test sequence will be as follows:

(1) Operate Hughes Engine

- (a) Steady state operation with spin rate change and beam diagnostic measurements. The neutralizer controls will be pre-set.
- (b) Start Hughes alternate programmer to vary electrode voltages. Provisions are made to stop the programmer at any point to measure spin rate change.
- (c) Switch to alternate neutralization method with no voltage on trap electrode. Spin rate change and beam diagnostic measurements will be made.

(2) Operate Lewis Engine

- (a) Steady state operation with measurement of spin rate change.

- (b) Probe surveys to measure the distribution of energy density about the beam center line.
- (c) A study of the response of the engine to step transients.
- (d) Probe surveys with neutralizer alternately on and off.

Effectively neutralized ion beams will be indicated if:

- (a) The thrust computed from payload spin rate data agrees with the theoretical thrust for the engines.
- (b) The measured beam spreading rates are in agreement with those obtained in vacuum tank tests.
- (c) The ion beam current and the various electrode drain currents are in agreement with vacuum tank data. In particular, the presence of beam turnaround should manifest itself by large currents to the accelerator and electron trap electrodes.

SERT II

The SERT II flight tests will consist of two Thor-Agena orbiting flights, and they will begin during the approximate time period indicated in figure 1. The engines to be tested on these flights will be modular units of the ion engines being developed for use with SNAP-8 and the larger power generating systems which will follow. A single ion engine operating at 2 to 3 KW input power will be tested in each of the flights. A contract for development of the payload package has not yet been awarded.

Purpose.- The objective of the SERT II flights is to make certain that the engine design concepts are suitable for space operation prior to going ahead with the expensive development and testing of larger engines. The operation of the neutralizer systems will again be an item of particular concern in these flights, since these neutralizer systems will be different in configuration than either of the systems tested in the SERT I flights. These flights should reveal any problems with the neutralization of closely-spaced multiple beams. Also, the perveance will be appreciably greater than in the SERT I engines.

The long-term effects on ion engine systems of micrometeoroids and corpuscular radiation can be accurately determined only by long-duration orbital flight tests with realistic engine geometry. Some possible effects are:

Triggering of electrical breakdown between the high-voltage electrodes,
and
Deposition of sputtered material on the ion emitter and on insulators.

Engines. - Only one engine is definitely scheduled for test in the SERT-II program. This engine is shown in figure 6. It is a strip-beam engine currently under development by Hughes Research Laboratories under NASA contract. It has the following performance goals:

- Overall frontal dimensions, 5 in. by 4 in.
- Engine weight, 12 pounds
- Propellant, cesium
- Specific impulse, 6000 sec.
- Total input power, 2.17 KW
- Power efficiency, 75 percent
- Propellant utilization efficiency, 98 percent
- Thrust, 12 millipounds
- Beam current, 685 MA
- Current density, 15 MA/CM²
- Beam perveance, 5.86 micropervs

The neutralizer system is shown in figure 6 and is similar to the one to be tested on the annular-beam engine in the SERT I flights.

A second engine which may be tested during the SERT II flights is a cesium electron bombardment engine which has recently began development under NASA contract at Electro-Optical Systems. The contract awarded December 27, 1962, calls for a development of a 3 KW engine suitable for clustering or scaling to larger sizes. Delivery of two flight prototype systems, including thrusters, propellant feed systems, and controls, is scheduled for one year from the contract award date.

Test technique. - Solar cells and rechargeable batteries will be used to permit continuous operation of a 2-3 KW engine. Engine thrust will be obtained from measured changes in the satellite orbit.

SERT III

The SERT III program is comprised of two Thor-Delta orbiting flights during the time period indicated in figure 1.

Purpose. - The purpose of the SERT III flights is to qualify an ion rocket system for mission application for attitude control and station-keeping of synchronous, stable-platform satellites. The Thor-Delta flights will place the ion engine system in low-altitude polar orbit and will serve to demonstrate the satisfactory operation of all system components: thrusters, gimbals, controls, propellant system, power supply, and attitude sensors. Of particular interest will be the long-term effects on system performance of such factors as micro-meteoroids, corpuscular radiation, hard vacuum, and zero gravity.

Engine system. - A contract was awarded Hughes Research Laboratories on October 1, 1962 to develop the ion engine system. A flight prototype system will be delivered one year from contract award date. The system (fig. 7) will contain eleven thrusters grouped into four pods. Each pod will contain two strip-beam, 0.5 millipound thrusters for attitude control. Three of the four pods will also contain an annular-beam, 1.5-millipound engine for station-keeping. Gimbals like those under development for the Surveyor project will be used to rotate each pod through 90 degrees, thus permitting each pair of attitude control engines to be used for attitude control about either of two major axes.

The porous tungsten ion emitter is warmed up by resistance heaters over a 30 second period prior to operation of each attitude thruster. A valve in the propellant supply line to each engine is used to initiate and to terminate the thrust. If finer thrust control proves desirable, it can be obtained by controlling voltage on the accelerator electrode.

On-board sensors will be used to supply attitude control signals, and station-keeping operation will be triggered by ground command. The control system will permit simultaneous attitude and station corrections.

The engine system, exclusive of solar cells and rechargeable batteries, is expected to weigh 47 pounds. This weight includes the propellant necessary for 3 years of operation on a 550-pound synchronous, stable-platform satellite.

Test technique. - The spacecraft attitude will be controlled to within ± 0.5 degrees in order to evaluate the attitude control system. The station-keeping thrusters will be operated at intervals. Some additional attitude disturbance torque is to be expected during this operation if the station-keeping thrust vectors do not pass through the spacecraft center of gravity. Uncertainties in c.g. location, thermal expansion effects on c.g. location, and thruster alignment errors are some of the causes of attitude disturbance torques during operation of the station-keeping thrusters. An evaluation of the magnitude of this effect is one of the major objectives of the SERT III program.

Zero-g Flights

A flight program, using Aerobee vehicles, is under way at the NASA Lewis Research Center to study zero-gravity heat transfer to fluids. Data from these zero-g experiments will provide assurance that the propellant feed systems will function properly in space. If it is deemed necessary, flights will be conducted as a part of the Aerobee zero-g program to test the specific propellant systems to be used in the various SERT flights.

EARLY ELECTRIC PROPULSION MISSIONS

Satellite Attitude Control and Station-Keeping

The mission. - The earliest mission application of electric propulsion is likely to be for attitude control and station-keeping of synchronous, stable-platform satellites. This application is considered sufficiently promising to warrant the immediate development of the ion rocket system described under the SERT III program even though no specific satellite has yet been scheduled to use the system. The present schedule of SERT III flight tests would provide us with a mission-ready system in calendar year 1965.

The attitude control system must be capable of overcoming the effects of disturbance torques which arise from natural effects such as solar radiation pressure and micrometeoroid impact and internal effects such as gas leakage and moving parts. The disturbance torques are a function of satellite configuration. The unbalanced solar pressure is generally the main disturbing torque, and for a carefully designed 550-pound satellite, this torque is likely to be of the order of 50 dyne-cm.

To maintain a satellite at a fixed position in a 24-hour orbit, the thrust requirements are dictated by three principal disturbing forces. The largest of these is due to the so-called triaxiality of the earth (non-symmetrical mass distribution about the polar axis). The other principal disturbing forces result from the gravitational attraction of the sun and the moon.

The triaxiality of the earth causes the satellite to drift in an Easterly or a Westerly direction toward the closer of two positions of minimum gravitational attraction. The rate of drift can be as great as 0.3 degree of longitude per day, depending upon the satellite's initial longitude.

The gravitational attractions of the sun and the moon cause the satellite orbit inclination to vary from zero to about 16 degrees in cyclical fashion and with a period of about 50 years.

For 3-axis attitude control and station-keeping for a stable-platform satellite it is necessary to apply corrective thrust in various directions. Attitude control about each axis requires two equal-magnitude thrust vectors, oppositely directed, and with at least one of these vectors providing torque about the axis. Station-keeping requires single vectors through the satellite center of gravity, in both the North-South and the East-West directions. These requirements can be met by a number of fixed thrusters, or by as few as two thrusters if they are suitably gimballed.

The system. - The system selected for development is shown in figure 7 and is described briefly under the SERT III program. It will weigh about 47 pounds, which includes propellant for 3 years of operation on a 550-pound

synchronous, stable-platform satellite. The system will contain 11 thrusters and 4 gimbals. These will provide 100 percent thruster redundancy for attitude control about all 3 axes and for station-keeping in the North-South direction. Many of the components of the system have already been developed for other applications, and selection of these components provides reasonable assurance of a satisfactory system at an early date. Delivery of the first prototype system is scheduled for October 1, 1963.

Comparison with chemical rockets. - It is interesting to compare the capabilities of ion rockets and chemical rockets for satellite attitude control and station keeping. This comparison will be made for a 550-pound synchronous satellite, a 3-year operating life, a stable platform design. If the satellite were spin stabilized as with Syncoms I and II, then electric propulsion would not be feasible. However, stable-platform satellites are being considered for missions beyond Syncom II. Stable platforms permit the use of fixed antennae as opposed to the phased antennae arrays of spinning satellites. Also, the attitude sensors and control system are simplified for a non-spinning satellite.

The following table compares the ion rocket system currently under development with two bi-propellant chemical rocket systems operating at 300 seconds specific impulse.

COMPARISON OF SYSTEMS FOR ATTITUDE CONTROL
AND STATION-KEEPING

Synchronous, Stable-Platform, 550-lb Satellite

	Ion rocket	Bi-propellant	
		0.01 Lb sec/pulse	0.003138 Lb sec/pulse
Operating life, years	3	1	3
Propellant consumption, lb	2.3	280	68
Thruster and propellant system weight, lb	41	-	53
Solar cell weight, lb	19	0	0
Battery weight, lb	60	0	0
Total system weight, lb	122	350	121
Reference	7	8	9

The two columns for the bi-propellant rocket are for two assumed values of the minimum impulse bit per pulse. The 0.01 lb-sec bit is typical for

propellant valves currently undergoing use; this impulse bit results in exorbitant propellant consumption. The 0.003138 lb-sec bit may be representative of advanced valves. Such valves are not yet operational. This impulse bit yields a chemical system having a weight competitive with the electrical system for 3-year life. If these system weights prove to be attainable, then the choice between electrical and chemical systems would probably be determined by reliability considerations. For lifetimes greater than 3 years the electrical system would, of course, compare more favorably.

Orbital Mapping

A second mission in which electric propulsion may find early application is that of detailed mapping of the Earth's magnetic field and radiation belts. A satellite payload in initial low altitude polar orbit can make use of electric propulsion to spiral slowly outward to give a thorough mapping of the magnetic and radiation fields which rotate with the Earth.

A comparison of an arc jet and an ion rocket for this mission is shown in the following table:

COMPARISON OF FIELD MAPPING CAPABILITIES

Initial orbit: Polar, 300 n mi. Initial weight: 1130 lb

	I	II	III
	1.5 KW Arc Jet	3 KW Ion	3 KW Ion
Specific impulse, sec	1000	5000	5000
Thrustor efficiency, percent	50	70	70
Mission time, days	90	90	177
Scientific payload, lb	295	333	295
Final orbit, n. mi.	5800	2000	5200

Columns I and II compare the two thrusters on the basis of equal satellite raising times. The ion rocket is considerably inferior to the arc jet on this basis; it yields a slightly larger payload but cannot raise the satellite to an altitude approaching that attainable with the arc jet. Columns I and II compare the two thrusters on the basis of equal payload weights. Here the ion rocket approaches the raising capability of the arc jet, but requires twice the raising time. The conclusion, then, is the same as that arrived at by other studies (e.g., ref. 10); namely, the arc jet appears better suited for this particular mission. A small arc jet thruster, designed for operation in the 1-1.5 KW range, is presently being developed by Plasmadyne Corp. under NASA contract; a reliable thruster has not yet been

obtained. The ion engines being readied for SERT II could also be used for this satellite raising mission with the sacrifice in performance noted above. The resistance-heated electrothermal rocket (refs. 11 and 12) offers even more promise than the arc jet for this mission; however, it is in a very early stage of development.

The efficiencies listed for both the ion rocket and the arc jet in the foregoing table are somewhat above the levels currently being obtained; it is hoped that these efficiencies will be available with reliable and durable thrusters within 2 years.

CONCLUDING REMARKS

Flight tests are a necessary part of the development of electric thruster to mission-ready status. In 1963 we hope to establish the feasibility of neutralizing high-perveance ion beams in space. Following this, we expect to make orbiting flights with 2 to 3 KW modules of a contact ion engine and a bombardment ion engine, both of which can be clustered to yield the large ion engines required in the future. By the end of the decade we expect to have 30 KW, or larger, models of ion engines ready for possible flight tests with the SNAP-8 or other electric propulsion systems.

Two early missions appear attractive for electric thrusters with solar cells as the power source. One of these missions is station-keeping and attitude control of synchronous, stable-platform satellites. A system containing 11 small ion thrusters is presently under development, and will undergo flight tests in 1965.

The second early mission for electric thrusters is the raising of small, instrumented satellites to give detailed mapping of the Earth's magnetic field and radiation belts. A small electrothermal rocket appears most attractive for this application, but an ion rocket could also suffice if a suitable electrothermal rocket does not become available.

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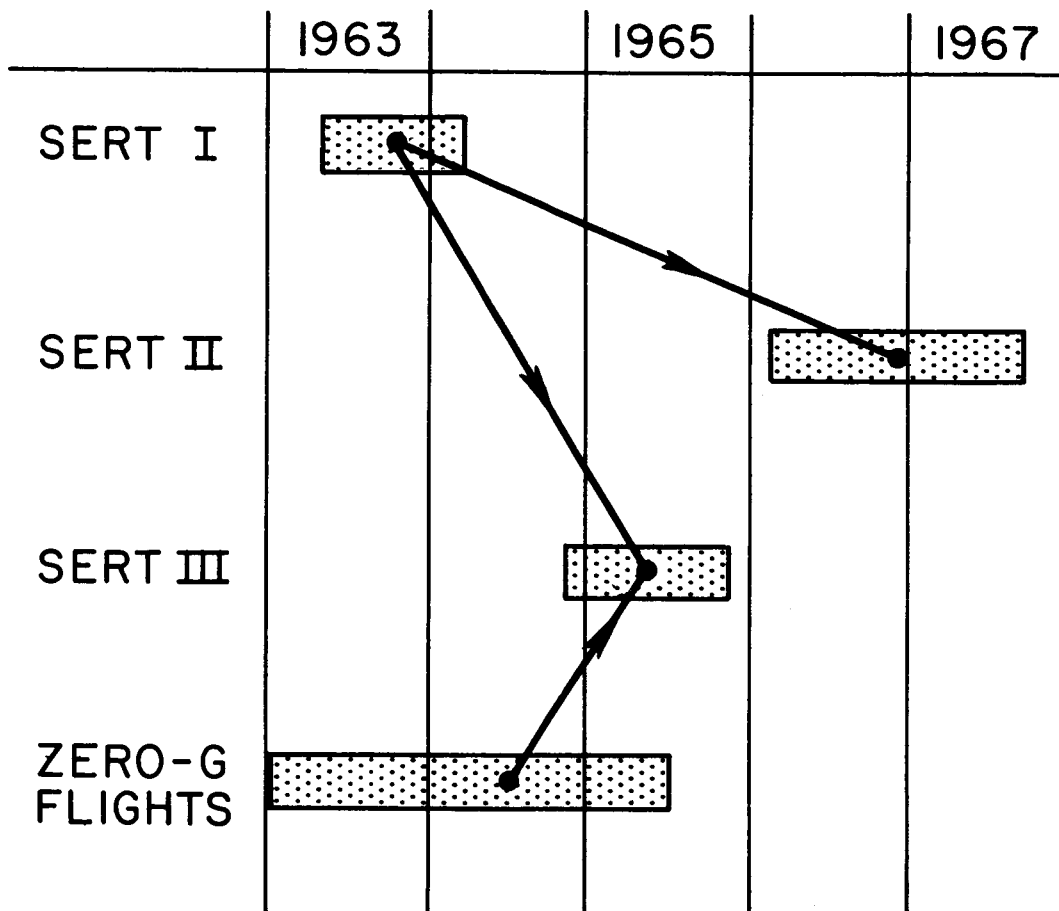


Fig. 1. - NASA electric propulsion flight tests.



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Fig. 3. - SERT I flight experiment.

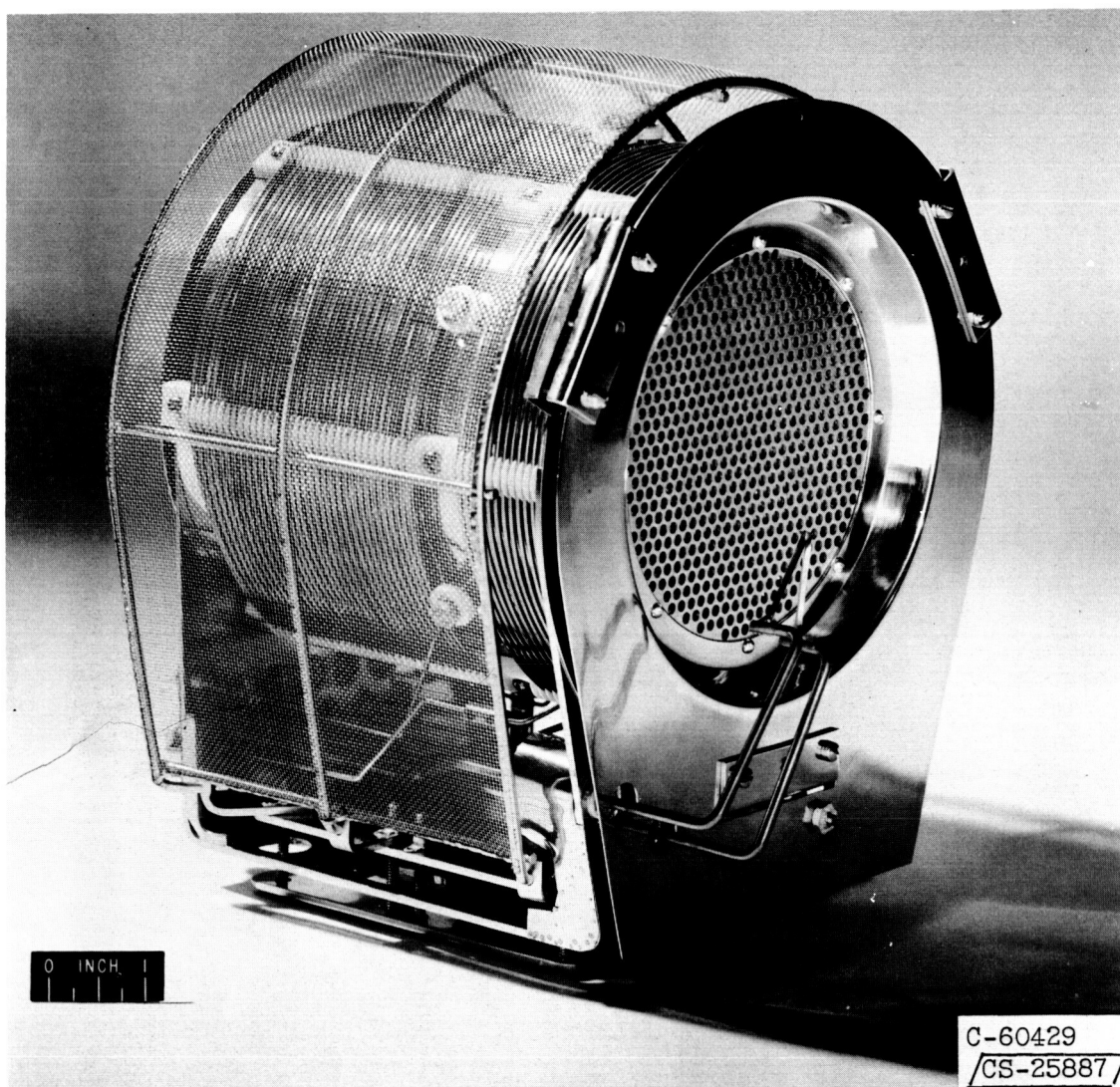


Fig. 4. - Lewis engine for SERT I.

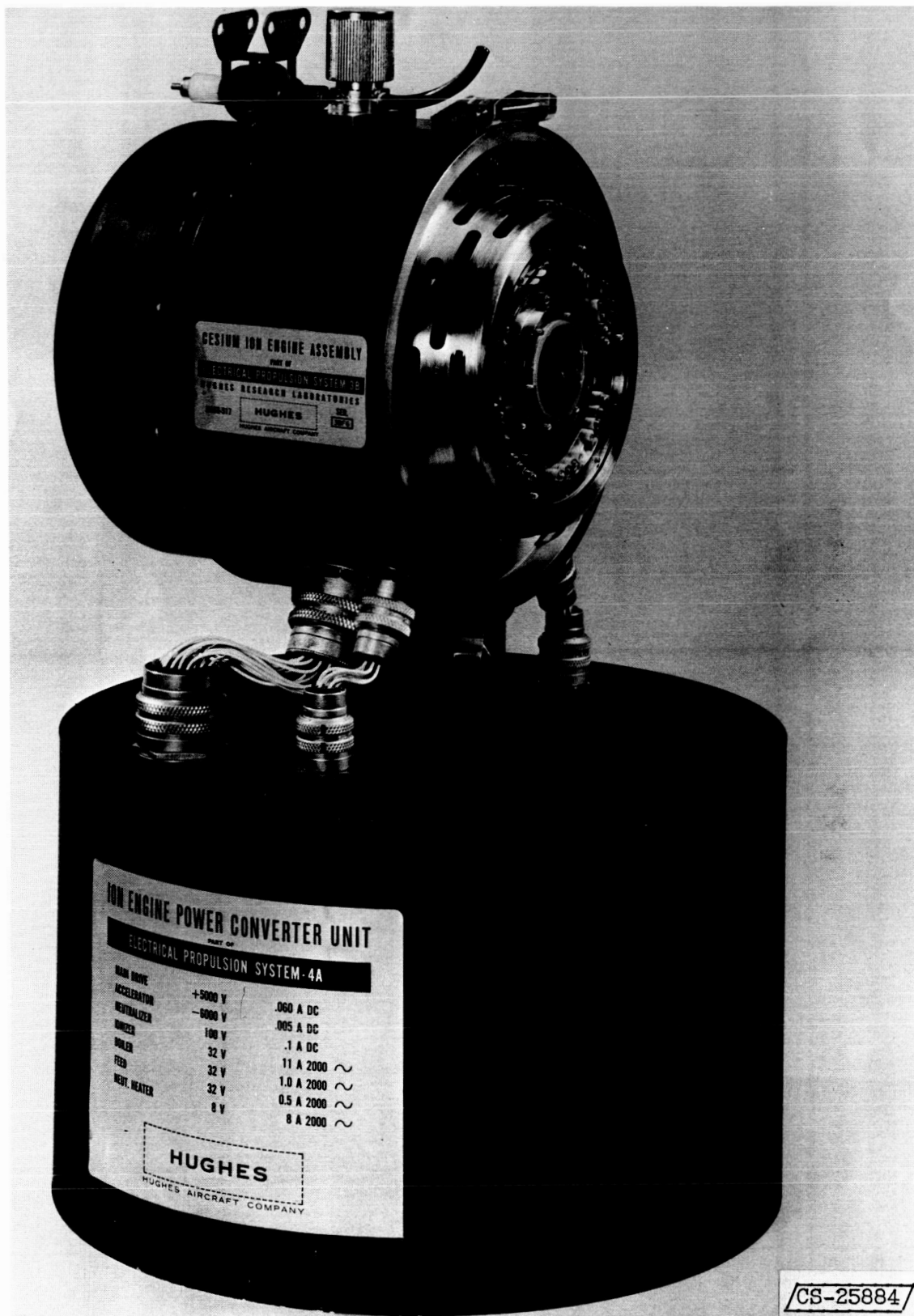


Fig. 5. - Hughes engine for SERT I.

HUGHES

PROPOSED DESIGN 0.010 LB LINEAR STRIP ION ENGINE

ACCELERATING ELECTRODE

IONIZER

CESIUM
MANIFOLD

FOCUS
ELECTRODE

NEUTRALIZER
& DECEL
ELECTRODES

THRUST = 0.0124 LB
POWER = 2.17 kw
I = 6000 sec
P/T = 175 kw/LB

CS-25885

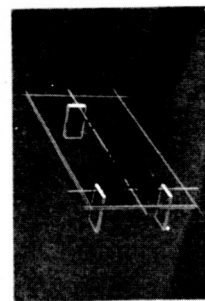


Fig. 6. - Hughes engine for SERT II.

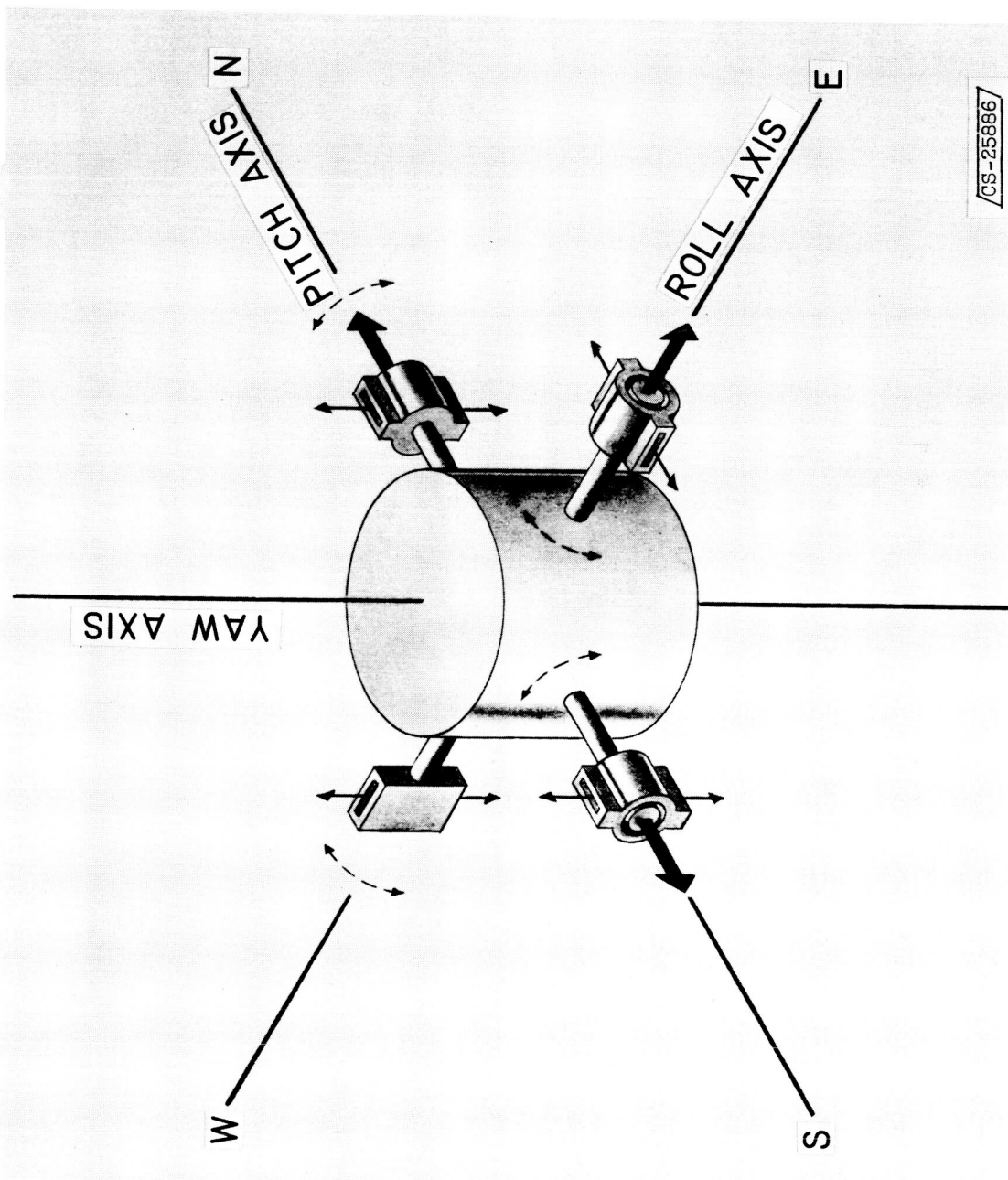


Fig. 7. - Ion engine system for satellite attitude control and station keeping.